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Space Programs Summary No. 37-38, Volume II

for the period January 1, 1966 to February 28, 1966

The Planetary–Interplanetary Program

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Preface

The *Space Programs Summary* is a six-volume, bimonthly publication that documents the current project activities and supporting research and advanced development efforts conducted or managed by JPL for the NASA space exploration programs. The titles of all volumes of the *Space Programs Summary* are:

Vol. I. The Lunar Program (Confidential)

Vol. II. The Planetary–Interplanetary Program (Confidential)

Vol. III. The Deep Space Network (Unclassified)

Vol. IV. Supporting Research and Advanced Development (Unclassified)

Vol. V. Supporting Research and Advanced Development (Confidential)

Vol. VI. Space Exploration Programs and Space Sciences (Unclassified)

The *Space Programs Summary,* Vol. VI consists of an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of technical supporting activities, including engineering development of environmental-test facilities, and quality assurance and reliability; and a reprint of the space science instrumentation studies of Vols. I and II.

W. H. Pickering, *Director*
Jet Propulsion Laboratory

Space Programs Summary No. 37-38, Volume II

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I. *Mariner* 1967 Project

A. Introduction

The *Mariner* 1967 Project continues and extends the work of the previous *Mariner* Projects. Three principal project areas include two which record the progress of the *Mariner IV* spacecraft now in flight; the third covers the development of a Venus flyby mission.

1. Mariner Mars 1964 Project

Residual activity of the *Mariner* Mars 1964 Project consists primarily of continuing analysis of tracking data, publication of results by the principal investigators, and preparation of final reports. However, periodic detection, recording, and commanding activities involving the *Mariner IV* spacecraft have been assigned to the Deep Space Network (DSN) of JPL. Using elements of the experimental facilities at Goldstone Station and advanced R&D techniques, the DSN will make attempts to communicate with the spacecraft. The result of this effort is published in *SPS*, Vol. III.

2. Mariner IV Project

In the second half of 1967, *Mariner IV* will again be within the normal communication range of the DSN stations. At that time an effort will be made to obtain additional telemetry data from *Mariner IV*. The *Mariner IV* Project comprises the preparation, support, and actual reacquisition of the spacecraft. Authorized simultaneously with the Venus flight effort at the end of December 1965, the objectives have been established as follows:

The primary objective of the *Mariner IV* Project is to obtain scientific information on the interplanetary environment in a region of space further from the Sun than the orbit of Earth during a period of increasing solar activity in 1967, using the *Mariner IV* spacecraft still operating in orbit around the Sun.

The secondary objectives are to obtain additional engineering knowledge about the consequences of

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extended exposure of spacecraft equipment in the interplanetary space environment and to acquire experience in the operation of a planetary spacecraft after a prolonged lifetime in deep space.

Flight operations on this project (providing a functioning spacecraft allows this to occur) will overlap the *Mariner* Venus 67 Project flight activities.

3. *Mariner Venus 67 Project*

The major task of the *Mariner* 1967 Project is the Venus flyby mission, known as the *Mariner* Venus 67 Project. The following mission objectives have been established for that effort:

> The primary objective of the *Mariner* Venus 67 Project is to conduct a flyby mission to Venus in 1967 in order to obtain scientific information which will complement and extend the results obtained by *Mariner II* relevant to determining the origin and nature of Venus and its environment.
>
> Secondary objectives are to acquire engineering experience in converting and operating a spacecraft designed for flight to Mars into one flown to Venus, and to obtain information on the interplanetary environment during a period of increasing solar activity.

This project was authorized late in December of 1965. With the launch opportunity occurring in June 1967, the period available to the *Mariner* Venus 67 Project for planning, design, test, and other flight preparation is 18 mo. The preparation period for *Mariner II* was 1 yr, and for *Mariner* Mars 1964 was 2 yr. This 18-mo interval between approval of the mission and launch opportunity requires application, to the fullest extent possible, of existing hardware and techniques and experience proved on prior missions.

The single flight spacecraft is currently being prepared by converting the spare spacecraft (*MC-4*)[1] that was built for the *Mariner* Mars 1964 project. An *Atlas-Agena D* launch vehicle will inject the spacecraft into a 120-day transit trajectory. Preliminary estimates set the launch window at approximately a 17-day period beginning in the middle of June 1967, with the encounter date tentatively set for mid-October, 1967.

Seven scientific experiments have been approved for the *Mariner* Venus 67 mission:

Experiment	*Principal investigator*
S-Band radio occultation	Arvydas J. Kliore, Jet Propulsion Laboratory
Ultraviolet photometer	Charles A. Barth, University of Colorado
Dual-frequency radio propagation	Von R. Eshleman, Stanford University
Helium magnetometer	Edward J. Smith, Jet Propulsion Laboratory
Solar plasma	Herbert S. Bridge, Massachusetts Institute of Technology
Trapped radiation	James A. Van Allen, State University of Iowa
Celestial mechanics	John D. Anderson, Jet Propulsion Laboratory

The first of these experiments requires the use of only the RF transmission subsystem on the spacecraft, and the last utilizes only the tracking doppler data derived from the RF carrier. Of the remaining five experiments, four are to be accomplished with the existing instrumentation, with only minor modifications. Only the remaining experiment, the dual-frequency radio propagation experiment of Stanford, requires incorporating a new scientific instrument into the payload.

Other changes to the basic spacecraft design are necessitated by the fact that the spacecraft will travel toward, rather than away from, the Sun and also because conversions must be made to accommodate revised encounter sequencing and science payload. In particular, modifications are needed in the following areas:

(1) Scientific data automation system

(2) Antenna pattern and orientation

(3) Thermal control

(4) Solar panel configuration

(5) Planetary sensors

[1]Portions of the *Mariner C* proof test model (PTM) spacecraft are also being prepared as a flight support spacecraft; there will be no proof test model as such. The flight support spacecraft will serve the double function of pseudo-PTM and backup spacecraft for qualifying spare subsystems.

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B. Engineering Mechanics

1. Introduction

The *Mariner* Venus 1967 Project requires the support of the following Engineering Mechanics in the following areas:

(1) Structure. Design and fabrication of the basic octagonal structure, superstructure, high-gain and low-gain antennas, and solar panels.

(2) Mechanical devices. Design and fabrication of solar panel and low-gain antenna dampers, cruise dampers and latches, panel deployment springs and latches, the separation-initiated timer and the pyrotechnic arming switch.

(3) Temperature control. The over-all temperature control of all spacecraft equipment and the louver, shield, and surface coatings required to maintain the spacecraft temperatures within limits.

(4) Electronic packaging. Hardware provisions for mounting the electronics within the spacecraft and consultation on the packaging of all spacecraft electronic items.

(5) Flight cabling. Design and fabrication subsystem interconnect harnesses, the ring harnesses, and all peripheral electrical harnesses.

Support must also be provided for the following systems test and operations involving the spacecraft system:

(1) Ground handling equipment. This includes all supporting, hoisting, positioning, and shipping hardware.

(2) OSE cabling. Operations support equipment interconnect cabling is provided for the systems test complex equipment, for environmental tests, and for launch complex equipment.

(3) Weight and center-of-gravity hardware. This hardware is used to determine experimentally the spacecraft center-of-gravity position and to position the post-injection propulsion system (PIPS) thrust through the spacecraft center of gravity at the midcourse maneuver.

Additional support is provided the project in the following engineering mechanics areas:

(1) Spacecraft materials and applications.

(2) Spacecraft structural and dynamic analysis and test.

(3) Electronic component procurement.

(4) Design, drafting, and specification documentation.

(5) Launch vehicle mechanical interface design, test, and hardware cognizance.

Subsequent issues of the *SPS* will contain reports of the significant developments on the *Mariner* Venus 1967 Project in the above areas.

A review of mechanical-configuration activities follows in Sect. 2. The *Mariner* Venus 1967 spacecraft will be mated to the booster and launched in the same orientation as *Mariner* Mars 1964. However, the design flight attitude has been "reversed" to properly orient the spacecraft antennas. The resulting changes to the spacecraft appendages, sensors, solar panels, and thermal shields are discussed.

The changes to the spacecraft thermal design associated with reversing the spacecraft solar attitude and the absolute changes in solar intensities to be encountered in flight to Venus are reviewed. The solar intensities vary from 130 w/ft^2 at launch to 250 w/ft^2 at Venus encounter. For *Mariner IV* the solar intensities ranged from 133 w/ft^2 at launch to 53 w/ft^2 at encounter. A complete series of space simulator tests are planned to develop and confirm the spacecraft thermal design.

Structure and dynamic analyses have been performed on the revised solar panel structural boost latching and damper design. These analyses confirm that the proposed design and configuration are feasible.

Shroud separation analysis on *Mariner* Mars 1964 made by JPL, supported by reports from Lockheed Missiles and Space Company on *Mariner* Mars 1964 and *Lunar Orbiter*, confirm that the shroud will not hit and damage the spacecraft during any possible nonstandard shroud-ejection sequence. Because of the results of the analysis and the relative complexity in adding guide rails similar to those used on *Mariner IV*, the requirement for guide rails has been deleted.

A new high-gain antenna structural support geometry will be required. A satisfactory superstructure design has evolved which can meet the currently defined constraints.

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2. Mechanical-Configuration Description

The *Mariner* Mars 1964 basic bus structure and packaging arrangement is utilized intact for the Venus mission. However, rearrangement of much of the attached equipment is necessary because of characteristics unique to the Venus trajectory and revisions in the science payload. The resultant configuration, as presently defined, is illustrated in Fig. 1.

a. Antennas. To satisfy Earth-pointing requirements, the omnidirectional and high-gain antennas are required to be oriented in the anti-Sun direction. In addition, the size of these antennas dictates that the under-shroud orientation cannot differ greatly from *Mariner* Mars 1964 without imposing serious design changes to the spacecraft and launch vehicle adapter. These restrictions thus made it necessary to physically locate the antennas as before by orienting the *Mariner* Venus 1967 spacecraft in-flight attitude reversed as compared to *Mariner* Mars 1964. Sun-oriented equipment such as Sun sensors, certain science experiments, and solar panels are then reoriented as required.

Specifically, the *Mariner* Mars 1964 low-gain antenna and all support bracketry are utilized without change. The high-gain support structure is modified to provide proper pointing direction and mechanization to change the pointing direction of this antenna by discrete steps for increased coverage during occultation and after Venus encounter.

b. Solar panels. Reversed Sun-line attitude of the spacecraft requires the solar panel supporting spars to be mounted on the same hinges but with the cell panels mounted on the reverse side of the supporting spars and with the cells facing the opposite direction. The panels are of the same length as those of *Mariner* Mars 1964, but with a cell area of 40 ft^2, instead of the previous 70 ft^2. Maintaining the original panel length and locating the solar cell area away from the bus allows the use of *Mariner* Mars 1964 attitude-control gas systems, permits much of the existing tooling and fixtures, and serves to provide needed thermal decoupling of the panels from the bus.

Panel deployment springs and cruise dampers are unchanged. The boost latching and dampers are replaced with orthogonal pairs of dampers that tie the panels together at the tips. Four pinpullers, instead of the eight used in *Mariner* Mars 1964, are used to release the panels.

c. Attitude-control sensors. Reversing the Sun attitude requires that the primary and secondary Sun sensors be interchanged. Secondary sensors are mounted on the existing primary sensor pedestals, and the primary sensors are now mounted on pedestals of sufficient height above the Sun-side thermal shield to obtain an unobstructed view. To provide an improved field of view, the Canopus tracker is also relocated from the bottom ring of Bay VIII to a centerline location on the top ring of the same bay.

d. Science instruments. Relocation of interplanetary science requires only minor modifications to satisfy Sun-orientation requirements. The trapped radiation detector is mounted at the same location as that of *Mariner IV*, with the instrument itself reversed to provide proper orientation. The plasma probe is located on a solar panel spar adjacent to its previous Bay III mounting and achieves proper Sun orientation by solar panel deployment.

Planetary science, consisting of a UV photometer, and dual-frequency receiver, are mounted on adapter brackets attached to the top ring. The UV is located over Bay II at a cone angle of 90 deg, to allow a field of view compatible with the planet encounter geometry. The dual-frequency receiver, having no view requirements, is placed over Bay IV as a matter of mounting convenience.

e. Exterior thermal hardware. Shield design and temperature control louver hardware remain unchanged with the exception of the Sun-side thermal blanket. In order to prevent heat leaks at Venus intensities, a shield has been designed with a minimum of penetrations for Sun-side mounted equipment. The mechanization proposed to satisfy these thermal requirements consists of a deployable shade erected following launch vehicle separation. All hardware interfacing with the launch vehicle adapter, such as in-flight disconnect, separation feet, push-off pads, and separation timer, will be shielded from the solar input by this shade.

3. Temperature Control

a. Thermal design. The *Mariner* Venus 1967 temperature control subsystem is functionally similar to that of *Mariner* Mars 1964. Unchanged design features include:

(1) Solar (lower) shield to minimize heat input from Sun.

(2) Upper thermal shield to minimize vertical temperature gradients.

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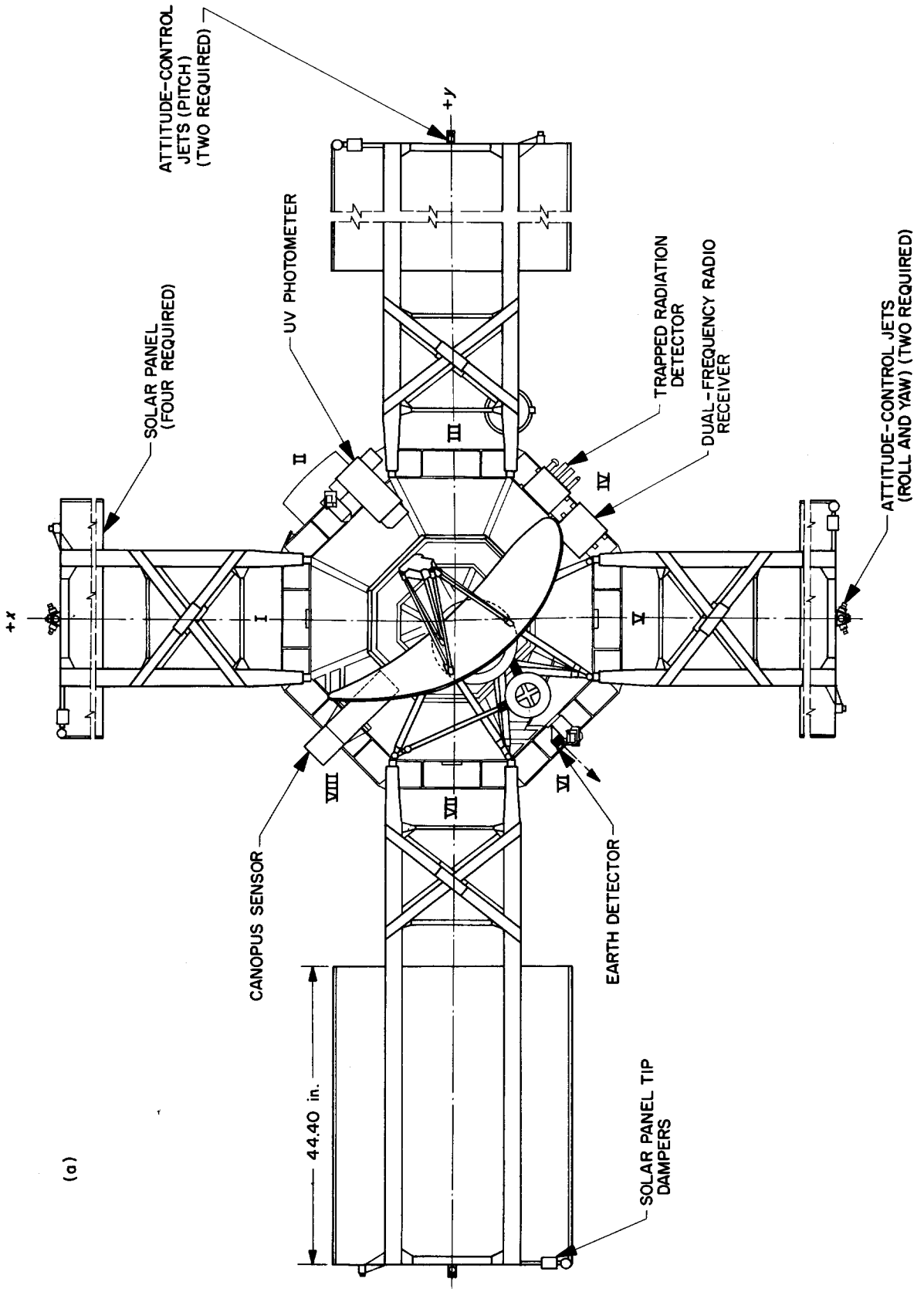

Fig. 1(a). *Mariner* Venus 1967 configuration, anti-Sun side

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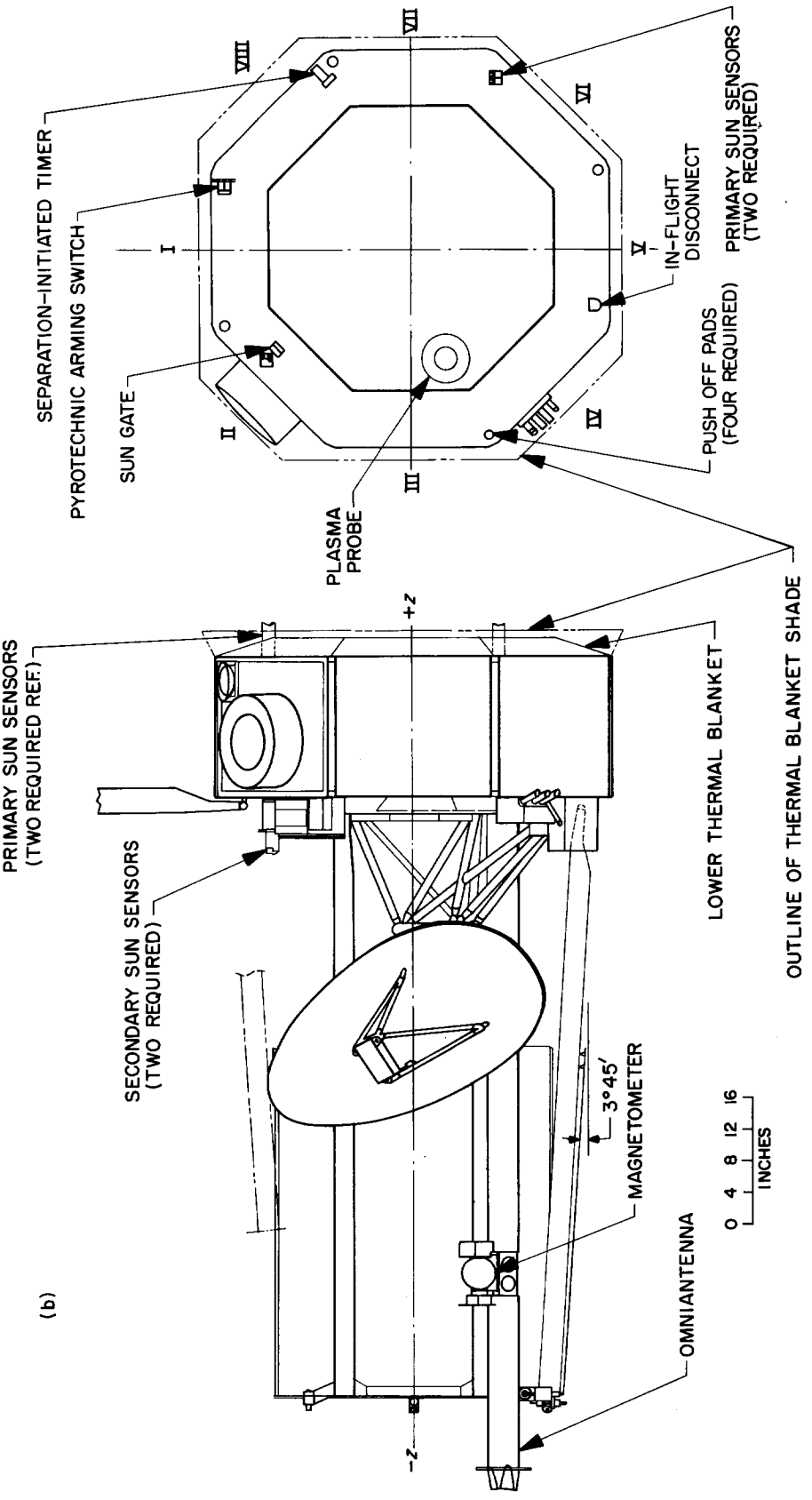

Fig. 1(b). *Mariner* Venus 1967 configuration, Sun side and Bay III side

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(3) Emittance control louvers on primary radiating surfaces (bay faces) to compensate for unavoidable variations in Sun input and internal power dissipation.

(4) High-emittance surfaces internal to bus to minimize temperature gradients.

(5) Science and appendages conductively and/or radiatively coupled with bus wherever possible.

(6) Isolated components passively controlled.

(7) Thermal design insensitive to solar simulator spectrum and imperfect collimation.

Some design changes are required, however, due to the nature of the mission and the reversed configuration. The solar intensity for this mission will be about 130 w/ft² at launch, 250 w/ft² at Venus encounter, and 455 w at perihelion. The *Mariner IV* solar intensity range was from 133 w/ft² to 53 w/ft². This extreme variation makes effective solar isolation an absolute requirement for spacecraft survival. The necessary improvements in isolation can be provided by two design changes. First, the solar input to the bus by irradiation from the solar panels can be reduced to *Mariner* Mars 1964 levels by eliminating 20 in. of cell area from the inboard ends of the panels. Second, the efficiency of the solar shield can be greatly improved by deploying a Sun shade over known leaks.

The reversal of the solar vector permits shade for most of the science and appendages. With the exception of the magnetometer and the antennas, these components can be conductively coupled to the bus. The magnetometer will require about 2 w of electrical heating to remain above the lower temperature limit. The antennas may be subjected to temperatures as low as −300°F, which could present structural problems. Spacecraft survival at perihelion is doubtful, because Sun sensors, attitude-control jets, and solar panels will be near their upper temperature limits at that time. Further design studies and firm temperature-limit definition are required for these components.

b. Temperature-control hardware. Relatively minor changes to louvers and peripheral shields are expected. The upper and lower superinsulation shields must be reconfigured to conform to structural changes. A feasibility model Sun shade has been fabricated and tested for proper deployment at spacecraft–Agena separation. Further development of this concept is under way.

Sunlit white paints will be changed from PV-100 and ARF-2, used on *Mariner* Mars 1964, to an S-13 type.

The materials used in the *Mariner* Mars 1964 absorptivity standard cannot withstand the elevated temperatures of a Venus mission, so this device has been eliminated from the *Mariner* Venus 1967 spacecraft.

c. Test plans. Space simulator tests with a rebuilt temperature-control model (TCM) are planned for June and August in the JPL 10-ft space simulator. The earlier test will be of the relatively crude parametric study type, while the latter test will provide confirmation of the adequacy of a firm thermal design. A chamber calibration and checkout will be carried out before these TCM tests. Thermal tests of the flight spacecraft will be performed after systems tests in the space simulator. Other developmental tests in small vacuum chambers will be performed, as necessary, to support analysis.

4. Structure and Dynamics

Static and dynamic structural analyses of major spacecraft components are utilized during a spacecraft program from preliminary design to test evaluation. Results from early analyses are used to establish structural configuration and to provide load estimates from which more detailed analyses of structural components and elements can be made. Analyses using a more refined structural idealization are then made to obtain analytical predictions, which are compared to, and updated by, model test results. Improved response analyses then provide a basis for evaluating test results and for determining the effects of any structural modifications.

a. Solar panel structural support configuration. The forced-response solution of the tip-latched *Mariner* Venus 1967 solar panels was initiated by performing a forced-response analysis of four tip-latched *Mariner* Mars 1964 panels, to verify the structural integrity of the proposed configuration.

The model was developed with each panel represented by an out-of-plane analysis of the panel idealized as a planar structure. The panel idealization, originally developed for *Mariner* Mars 1964, was modified by moving the damper attachment points to the panel top corners. The panels were interconnected by spring dampers, and the in-plane support was represented by a spring chosen to allow the in-plane rigid body frequency to match *Mariner* Mars 1964 test data.

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The composite analysis retained six out-of-plane modes and one in-plane mode for each panel, and eight modes corresponding to the dampers, making a total of 36 elastic modes. Modal damping of 1.0% critical was used for all normal modes. The constraint modes of the panels and all rigid body modes, except that of the base, were independent modes.

There were two sets of 18 undamped eigenvalues with associated eigenvectors that were completely decoupled. (This was true only because the in-plane loads on the panels were applied by the dampers at the shear center.) Because the two sets had corresponding frequencies and mode shapes, the problem was reduced to the solution of one 18-degree-of-freedom system.

A second configuration was analyzed which had only one out-of-plane damper to each panel. Fig. 2 compares the response of a point 42 in. up from the bottom on the light side of a panel. The input is a unit acceleration normal to the panel. The point chosen is near critical in each case.

Results of the preliminary analysis indicate that the proposed tip-latch configuration is reasonable, and effort is continuing to improve the in-plane model, to evaluate the true in-plane model damping, the effect of a change in damping coefficients, and the effect of the in-plane load not at the shear center.

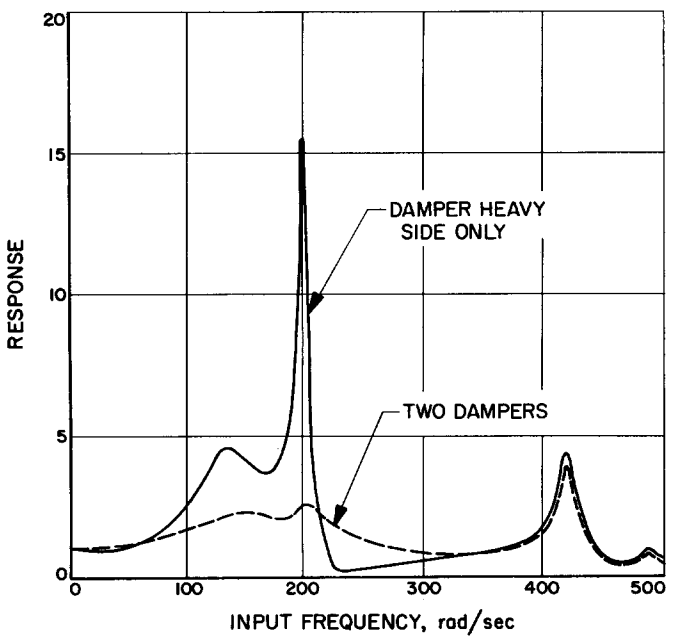

Fig. 2. Dynamic response of damped solar panels

b. Shroud separation dynamics. The problem of a possible collision between the *Mariner* Venus 1967 shroud and spacecraft due to one nonfunctional shroud separation spring bank has been analytically investigated to aid in the definition of the requirements for solar panel protective devices.

Two cases were considered. For Case 1, the manufacturing and assembly tolerances (i.e., spring-stroke, preload, stiffness, alignment, and center-of-mass offset) and the maximum residual vehicle rate were applied in a manner which was considered to maximize the chance for the shroud aft ring to collide with the solar panels. For Case 2, these parameters were applied in a manner which was considered to maximize the chance for a collision between a solar panel tip and an intermediate point on the shroud.

The JPL-developed Separation Dynamics Computer Program was used to carry out the required dynamic analyses, and hand calculations served to translate the computer output into useful shroud–spacecraft relative position data. The spacecraft envelope was conservatively taken to be an axisymmetric figure whose radius at any axial station is equal to the maximum spacecraft radius at that station.

For both cases, it was determined that at least 1 in. of clearance remained. Therefore, no analytic justification for solar panel protective devices to prevent damage due to the "one bank out" condition was obtained from the analysis.

c. High-gain antenna structural support. Initial program guidelines called for relocating the high-gain antenna. The resulting geometrical arrangement requires that the superstructure be redesigned. Reorientation of the high-gain antenna and the requirement for a two- or three-position antenna necessitated offsetting the antenna support ring from the center of the bus. Static and dynamic structural analyses were performed for several design configurations.

For a preliminary design, a motor mount tubular truss from four adjacent bus points to the offset antenna support ring was analyzed and found marginally stable. Different modifications of this preliminary design were considered. The most desirable design required the antenna support ring to be raised 2 in. to allow the addition of two more members to the superstructure to increase the overall stiffness.

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C. Guidance and Control

1. Spacecraft Power

The use of the *Mariner* Mars 1964 power system for the *Mariner* Venus 1967 spacecraft requires that several modifications be made to the power system. The most important constraint to the spacecraft is that it must fly toward rather than away from the Sun; for the power system this constraint has the greatest effect on the solar panels. Changes to the power system are to be made, based on either new system requirements or on design weaknesses within the power system. New system requirements that cause changes are: (1) that the cruise science instruments are not to be turned off at the end of the encounter sequence, and (2) that the spacecraft will require more power than on the *Mariner* Mars 1964 mission. The other modification is a change in the battery voltage telemetry monitor to prevent it from trickle charging the battery.

On the *Mariner* Mars 1964 mission, with the spacecraft going away from the Sun, both the insolation available to the panels and the temperature of the panels decreased with time. The characteristics of the solar cells are such that the power output decreases as expected, but the output voltage increases with the decreasing temperature. On the Venus mission the power available from the arrays will be increasing with flight time, but the voltage will be decreasing. Since the size of the *Mariner* Mars 1964 array was determined by the power required at Mars, the array can be smaller for the Venus mission for a given power requirement. The series-parallel arrangement of cells in each panel section must be revised, however, because the voltage of the *Mariner* Mars 1964 panels would be only 30 v at Venus encounter; this low a panel voltage would cause uncontrollable sharing between the panels and the battery.

A further consideration in the panel design is caused by the mounting location of the panels. Because of the geometry of an Earth–Venus mission, the spacecraft roll axis must be reversed. Consequently, the side of the panel which faces the Sun would now reflect heat into the electronic bays. This has been relieved by eliminating approximately the first 2 ft of substrate and solar cells next to the octagonal structure. Sufficient area remains in the four panels to provide the power required near Earth.

Each panel has been rearranged to provide three sections per panel, with each section having a matrix of 105 series cells by 14 parallel strings. This allows the encounter solar panel voltage to be around 41 v, well above a battery sharing condition. The near-Earth panel voltage would be 55 to 56 v in the absence of the zener diodes, which limit this voltage to about 49 v. Four solar panels will be used to provide sufficient power. A two-panel configuration was considered but does not provide enough area to allow the cutout next to the spacecraft bus and still provide the near-Earth power required.

The change in turnoff procedure for the cruise science instruments is a result of their position on the *Mariner* Venus 1967 spacecraft and the reversal of the spacecraft roll axis. These instruments, which were in the Sun on the *Mariner* Mars 1964 mission, will be on the shaded side of the spacecraft during the flight to Venus. There is concern that if they are turned off after encounter, as now mechanized, they would become too cold and could not be returned to normal operation. The power to the cruise and encounter science equipment is controlled by the power distribution assembly of the power system. The switching logic for *Mariner* Mars 1964 allowed either the spacecraft CC&S MT-8 or MT-9 commands, as well as the DC-26 ground command, to turn off all equipment. Modification of the power distribution will prevent the MT-8 or MT-9 commands from turning off cruise equipment, but still allow its turnoff by the DC-26 ground command.

The increasing spacecraft power demand is caused by the revised encounter instrumentation and the required new design in the science data automation system (DAS) for the Venus mission. The power system main 2.4-kc inverter, which supplies the spacecraft subsystems, was designed for a maximum capability of 80 w continuously and 120 w peak. During the encounter sequence, between 90 and 120 w will be required on the Venus flight. The present inverter could not provide this power without degradation of the unit and reduction of the output voltage below the allowable lower limit. Two changes are being investigated to provide a continuous 120-w capability. The main transformer could be rewound with larger wire to decrease the internal resistance, and the output transistors could be paralleled to reduce the saturation resistance. The modification of the present units is not practical, because changes of this magnitude will require rearrangement of the parts within the unit. Fabrication of new assemblies will be required if the increased power capability is needed.

One other change is required in the power system. It was found that the mechanization of the battery voltage telemetry monitor allows a small trickle charge current to flow into the battery continuously. This may allow an

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overcharge condition to occur when the battery has been fully charged and the battery charger shut off. The *Mariner IV* battery continued to increase in terminal voltage beyond its cutoff point of 34.6 v after the charger was turned off, and in October 1965, when the spacecraft data ended, was at a terminal voltage of 37.2 v. Although the exact reason is not known, it is suspected that the battery has been charged to this voltage by the current through this telemetry monitor. If this is true, the battery voltage will continue to rise. The extent of damage that will occur to the battery under these circumstances is unknown, but rupture of a cell or cells is possible with the continued overcharging.

The proposed change would have replaced the present battery charger with one of a new design. This battery charger/booster provides two functions in the power system. It charges the spacecraft battery when required, but also provides a means of boosting out of an unnecessary sharing condition between the solar panels and battery. The circuitry used in the present design consists of a converter and magamp to provide the boosting, with the addition of a regulator on the output of the magamp to provide the battery charging. Although this mechanization provides satisfactory operation, the conflicting requirements of high voltage for boosting and lower voltage for charging required a change in the output voltage, according to the function needed. This was accomplished by switching a bias voltage on the magamp. The dual function required of the converter caused it to dissipate excessive power when charging and reduced its efficiency. The proposed battery charger/booster design separates the charging and boosting functions. This not only increases the efficiency in both modes, but also eliminates the dependency of one function on the circuitry for the other, increasing overall reliability. The proposed booster also incorporates a pulsed operation when boosting, with approximately a 10% duty cycle. This improves the booster efficiency and reliability, since it does not have to operate at full output continuously.

2. Attitude-Control Gas Storage

The attitude-control system for the *Mariner* Venus 1967 spacecraft is virtually identical to the *Mariner* Mars 1964 spacecraft. Therefore, the gas storage requirements for the *Mariner* Venus 1967 spacecraft will differ from the *Mariner* Mars 1964 spacecraft only as far as the spacecraft configuration and mission times differ. For example, different principal moments of inertia and gas jet moment arms change the gas usage rate. A shorter mission time to Venus will cause less usage due to solar torques and leakage. Therefore, a parametric study is made to determine the effect of various spacecraft configurations and various storage weights at launch.

There are three different ways in which spacecraft configuration and mission time can influence gas usage:

(1) Gas usage that is dependent only upon spacecraft configuration. This would include usage for the following functions: reduction of spacecraft–*Agena* separation rates to the controlled acquisition rates; acquisition transients and searches; commanded turns for midcourse correction; and interaction with autopilot during midcourse motor burn.

(2) Gas usage that depends upon both spacecraft configuration and mission time. Cruise gas usage under the influence of solar torques is in this category.

(3) Gas usage that is dependent only upon mission time. The normal leakage from the gas jets is in this category.

The various modes of spacecraft operation described above are the only ones in which gas storage requirements are significant. Hence, they are used to make the parametric study mentioned earlier.

a. Spacecraft configuration gas system requirements. The output of the gas jets is a constant control torque acceleration α, requiring constant thrust whenever the gas jets are on. Therefore

$$F = \frac{J}{L}\alpha \tag{1}$$

where

F = gas jet thrust

J = principal moment of inertia about the controlled axis

L = moment arm of gas jet

The thrust from the gas jet is

$$F = \dot{M} I_s \tag{2}$$

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where

$\dot{M}$ = weight flow of gas

I_s = specific impulse of gas

Then it may be seen that when the gas jets are on, the rate of gas usage is

$$M = \frac{J}{L}\frac{\alpha}{I_s} \tag{3}$$

Therefore, the spacecraft configuration gas usage may be quantitatively defined in terms of (J/L).

b. Redundant gas system requirements. The spacecraft has two independent gas systems. The gas jets are built in couples, so that there are two jets for each direction of rotation. Each of the two jets is plumbed to a different gas bottle. If a failure occurs, and all the gas stored in one bottle is depleted, one uncoupled jet remains to supply control torques, providing half the normal control acceleration.

In the event that a failure is caused by a valve sticking open, the torques thrown into the spacecraft by the failed jet will be cancelled by the jets of opposite polarity. One of the opposing jets is plumbed to the same bottle as the failed jet. The net result is that one third of the gas in the nonfailed bottle will be depleted, in compensating for the failed system. After a valve-open failure, the total gas remaining will be one third of the total gas before the failure.

c. Gas usage dependent on configuration only. As stated previously, there is a class of gas usage dependent on spacecraft configuration only. The contributing factors are analyzed as follows:

Separation rate removal. The total valve "on time" t_{on} is

$$t_{on} = \frac{\dot{\theta}_i}{\alpha}$$

where

$\dot{\theta}_i$ = the separation rate about a controlled axis

Then from Eq. (3) the weight of gas used for one axis is

$$M = \frac{J}{L}\frac{\alpha}{I_s}t_{on} = \frac{J}{L}\frac{\dot{\theta}_i}{I_s}$$

For all three axes, if 10% is added for misalignment effects, the total gas used for removal of separation rates of second-stage booster is

$$M = 1.1\left(\frac{J}{L}\right)\frac{\dot{\theta}_i}{I_s}$$

where

$$\left(\frac{J}{L}\right) = \left[\frac{J_x}{L_x} + \frac{J_y}{L_y} + \frac{J_z}{L_z}\right]$$

and x, y, z refer to the pitch, yaw, and roll axes, respectively.

Acquisition transients and roll searches. The controlled acquisition rate, determined by the Sun sensor saturation and gyro rate feedback is θ_a. For an overdamped system (high rate to position gain), the gas usage is determined as for separation rate removal. Three acquisitions are considered: (1) initial acquisition, (2) after midcourse, and (3) one spare. Therefore

$$M = 3\Sigma\left(\frac{J}{L}\right)\frac{\dot{\theta}_a}{I_s}$$

For roll searches and roll override operations, gas usage is proportional to how much angular rate is added and removed for each operation. Ten such operations are allowed for. Then,

$$M = \frac{J_z}{L_z}\frac{\Delta\dot{\theta}}{I_s}$$

where z refers to the roll axis, and $\Delta\dot{\theta}$ is the total accumulated rate change.

Commanded turns. The gas used for commanded turns is due to starting and stopping transients. The "on time" of the jets must be determined from control system analysis. Two complete maneuvers are allowed for, each being a pitch-roll sequence. Then for a normal, nonfailed system

$$M = 2\left[\frac{J_x}{L_x}t_x + \frac{J_z}{L_z}t_z\right]\frac{\alpha}{I_s}$$

For a failed system, where α is cut in half,

$$M = \left[\frac{J_x}{L_x}t_x + 2\frac{J_z}{L_z}t_z\right]\frac{\alpha}{I_s}$$

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where

t_x, t_z = transient "on times" of the pitch and roll valves, respectively.

Autopilot interaction. If the autopilot develops rather large oscillations, in the worst case, the attitude-control system could continuously consume gas. For a normal system

$$M = \Sigma\left(\frac{J}{L}\right)\frac{\alpha}{I_s}t_B$$

For a failed system (half α)

$$M = \tfrac{1}{2}\Sigma\left(\frac{J}{L}\right)\frac{\alpha}{I_s}t_B$$

where t_B = motor burn time.

d. Gas usage dependent on both configuration and mission time. During cruise, the percentage of time the valves are on to remove disturbance torques is proportional to the ratio of the disturbance torque acceleration to control torque acceleration.

$$\text{Total "on time"} = \frac{\alpha_d}{\alpha}t_l$$

where

α_d = disturbance torque acceleration

t_l = total mission time.

Then

$$M = \left[\frac{J_x}{L_x}\alpha_{dx} + \frac{J_y}{L_y}\alpha_{dy} + \frac{J_z}{L_z}\alpha_{dz}\right]\frac{t_l}{I_s}$$

e. Gas usage dependent on mission time only. Gas valve leakage is dependent on valve mechanization and must conform to specification, so that for a normal system

$$M = \dot{m}_l\, t_l$$

And for a failed system

$$M = \tfrac{1}{2}\,\dot{m}_l\, t_l$$

where

$\dot{m}$ = rate of gas usage, lb/sec

$\dot{m}_l$ = rate of gas usage during t_l

f. Total gas usage. The total gas usage for a full normal system may be summarized as follows:

$$M = \Sigma\left(\frac{J}{L}\right)[1.1\,\dot{\theta}_i + 3\,\dot{\theta}_a + \alpha t_B]\frac{1}{I_s} + \left[2\frac{J_x}{L_x}t_x + 2\frac{J_z}{L_z}t_z + \frac{J_z}{L_z}\frac{\Delta\dot{\theta}}{\alpha}\right]\frac{\alpha}{I_s} + \left[\frac{J_x}{L_x}\alpha_{dx} + \frac{J_y}{L_y}\alpha_{dy} + \frac{J_z}{L_z}\alpha_{dz}\right]\frac{t_l}{I_s} + \dot{m}_l\, t_l \qquad (4)$$

For purposes of determining the effect that spacecraft configuration has upon gas usage, it would be very convenient to be able to factor out $\Sigma\,(J/L)$. But, as may be seen, this is not directly possible. However, a useful approximation may be made. The second and third terms in Eq. (4) may be exactly determined for a particular spacecraft, in this case, *Mariner IV*. Then these terms may be defined as follows:

$$\left[2\frac{J_x}{L_x}t_x + 2\frac{J_z}{L_z}t_z + \frac{J_z}{L_z}\frac{\Delta\dot{\theta}}{\alpha}\right]\frac{\alpha}{I_s} \equiv c_1\,\Sigma\left(\frac{J}{L}\right)$$

$$\left[\frac{J_x}{L_x}\alpha_{dx} + \frac{J_y}{L_y}\alpha_{dy} + \frac{J_z}{L_z}\alpha_{dz}\right]\frac{t_l}{I_s} \equiv c_2\,\Sigma\left(\frac{J}{L}\right)t_l$$

where

$$\Sigma\left(\frac{J}{L}\right) = \left[\frac{J_x}{L_x} + \frac{J_y}{L_y} + \frac{J_z}{L_z}\right] \text{ for } \textit{Mariner IV}$$

The numerical value of c_1 and c_2 are determined, and Eq. (4) may be rewritten as follows:

$$M = k_1\,\Sigma\left(\frac{J}{L}\right) + k_2\,\Sigma\left(\frac{J}{L}\right)t_l + \dot{m}_l\, t_l \qquad (5)$$

Thus the three different classes of gas usage are shown, "normalized" about the *Mariner IV* configuration. It may be seen that when actual numbers are used, the approximated terms are not very significant.

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In a like manner, the gas usage for a failed (half) system is:

$$M(f) = k_3 \Sigma\left(\frac{J}{L}\right) + k_4 \Sigma\left(\frac{J}{L}\right) t_l + \tfrac{1}{2}\dot{m}_l t_l \qquad (6)$$

g. Failure mode analysis. Let M_0 be the initial gas weight stored at launch. Then if a failure occurs after launch, the gas remaining, M_R, is one third of the total remaining at failure time t_f.

From Eq. (5)

$$M_R = \tfrac{1}{3}\left[M_0 - k_1 \Sigma\left(\frac{J}{L}\right) - k_2 \Sigma\left(\frac{J}{L}\right) t_f - \dot{m}_l t_f\right] \qquad (7)$$

Then, from Eq. (6), the lifetime remaining in the system, t_R, will be

$$t_R = \frac{M_R - k_3 \Sigma\left(\frac{J}{L}\right)}{k_4 \Sigma\left(\frac{J}{L}\right) + \tfrac{1}{2}\dot{m}_l} \qquad (8)$$

Now it may be seen that the total lifetime of the system from launch, t_l, is

$$t_l = t_f + t_R \qquad (9)$$

h. Lifetime analysis. By the use of Eqs. (7), (8), and (9), two parametric plots are made. In order to show the effect of spacecraft configuration, linearized about the *Mariner IV* parameters, the total lifetime of the system from launch is plotted against the time of a possible

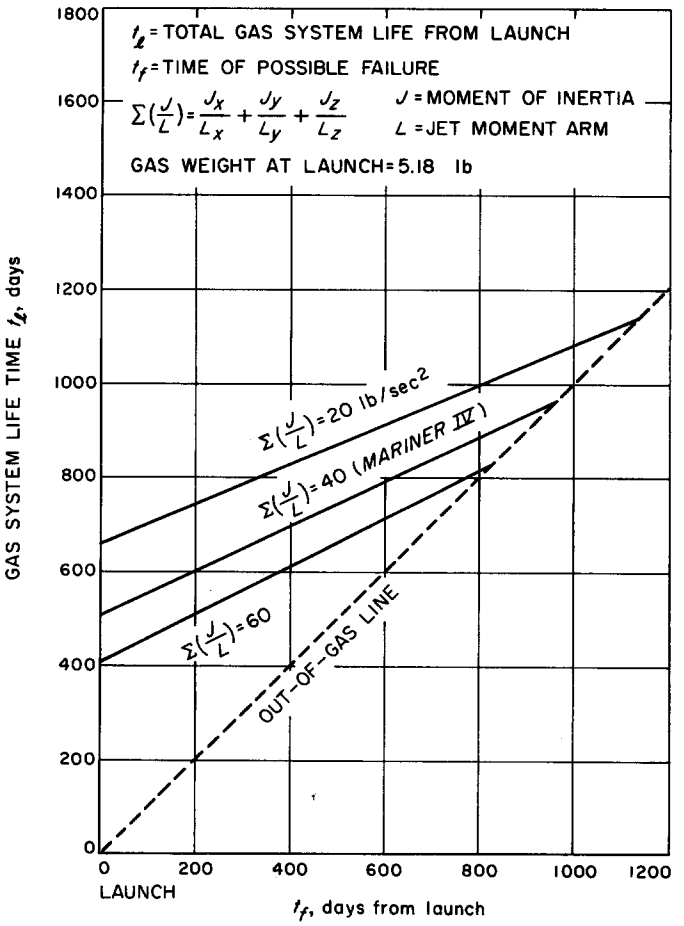

Fig. 3. *Mariner* Venus 1967 gas storage lifetime for given failure times after launch for various spacecraft configurations

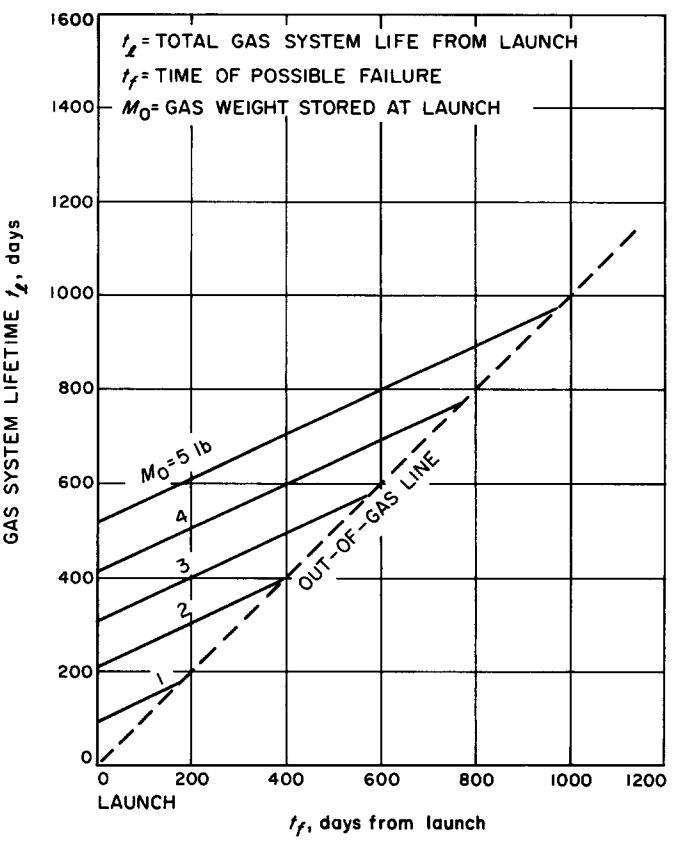

Fig. 4. *Mariner* Venus 1967 gas storage lifetime for given failure times after launch for various gas storage weights

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failure after launch, for various $\Sigma (J/L)$. The initial gas weight at launch, M_0, is 5.18 lb (for *Mariner IV*). This plot is Fig. 3.

Fig. 4 is a plot for the exact *Mariner IV* configuration, $\Sigma (J/L) = 40$ lb-sec^2, for various initial storage weights, M_0.

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II. *Voyager* Project

A. Introduction

1. Objectives

The primary objective of the *Voyager* Program is to carry out scientific investigations of the solar system by instrumented, unmanned spacecraft which will fly by, orbit, and/or land on the planets. Emphasis will be placed on acquisition of scientific information relevant to the origin and evolution of the solar system, the origin, evolution, and nature of life, and the application of this information to an understanding of terrestrial life. The primary objective of the *Voyager* mission to Mars, beginning in 1973, is to obtain information relative to the existence and nature of extraterrestrial life, the atmospheric, surface, and body characteristics of Mars, and the planetary environment by performing unmanned experiments on the surface of, and in orbit about, the planet. A secondary objective is to further our knowledge of the interplanetary medium between the planets Earth and Mars by obtaining scientific and engineering measurements while the spacecraft is in transit.

2. Project Plan

All *Voyager* missions will be conducted as events of an integrated program in which each individual flight forms a part of a logical sequence in an over-all technical plan of both lander and orbital operations. The *Voyager* design will provide for the carrying of large scientific payloads to the planet, the telemetering of a high volume of data back to Earth, and long useful lifetimes in orbit about the planet and/or on the planetary surface. Hardware will be designed to accommodate a variety of spacecraft and/or capsule science payloads, mission profiles, and trajectories. Particular emphasis will be given to simple and conservative design, redundancy wherever appropriate, and a comprehensive program of component, subsystem, and system testing.

Over-all direction and evaluation of the *Voyager* Project is the responsibility of the Office of Space Science and Applications (OSSA) of the National Aeronautics and Space Administration (NASA). Management of the *Voyager* Project and implementation of selected systems is the responsibility of the Jet Propulsion Laboratory (JPL) of the California Institute of Technology (CIT).

3. Technical Description

Two *Voyager* planetary vehicles are to be designed, constructed, and tested for launch on a single *Saturn V* during the 1973 Mars opportunity. Attention is also being given to requirements imposed on such vehicles by launches subsequent to 1973, such as a similar mission planned for 1975. Each planetary vehicle is to consist of a flight spacecraft and a flight capsule, with science experiments conducted in 1973 both from the orbiter and on the planetary surface. The flight spacecraft with its several hundred pounds of science payload will weigh approximately 2500 lb, its retropropulsion subsystem may additionally weigh up to 15,000 lb, and the flight capsule

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will weigh about 3000 lb. The flight spacecraft will be a fully attitude-stabilized device utilizing celestial references for the cruise phase, and will be capable of providing velocity increments for midcourse trajectory corrections and for Mars orbit attainment by both the flight spacecraft and flight capsule. Onboard sequencing and logic will be provided, as well as ground command capability. The flight spacecraft will supply its own power from solar energy or from internal sources and will be capable of maintaining radio communications with Earth. In addition, the flight spacecraft will be thermally integrated and stabilized and will monitor various scientific phenomena near Mars and during transit, and telemeter this information back to Earth; it will also monitor and telemeter data pertaining to spacecraft operation. The flight spacecraft will also provide the flight capsule with services such as power, timing and sequencing, telemetry, and command during the transit portion of the missions and may also serve as a communications relay. The sterilized flight capsule will be designed for separation from the flight spacecraft in orbit, for attaining a Mars impact trajectory, for entry into the Mars atmosphere, descent to the surface, and impact survival, and for surface lifetimes of two days in 1973 and later goals of as much as 6 mo. The flight capsule will contain the power, guidance, control, communications, and data handling systems necessary to complete its mission.

No deep space-flight tests of the flight spacecraft are planned, and compensation will be made by more extensive ground tests. The test program for the flight capsule, which is planned to include Earth-entry flight tests, will investigate entry dynamics.

B. Engineering Mechanics

1. Temperature Gradients Within Component Parts When Exposed to Heat Sterilization

a. Introduction. The present heat-sterilization requirements for *Voyager* specify that applicable hardware be exposed to specific microbe decontamination temperature–time cycles. There is a definite microbe decontamination level versus temperature–time relationship. The problem is to determine the temperature–time relationship within sealed component parts versus time–temperature ambient exposure; i.e., to determine if the interior of component parts actually is subjected to the full time–temperature conditions required for specific microbe decontamination levels.

b. Objective. The temperature-gradient phase of the heat-sterilization test program was conducted to determine the temperature gradient between the exterior and interior of the component parts and the length of time for internal temperature stabilization of component parts when placed in an ambient temperature of 135°C (simulated sterilization dry heat).

c. Program description. Two each of five component part types were selected for test. Temperature sensors were located in the interior and on the exterior of each part. Each part was individually inserted into a temperature chamber, preheated to 135°C, with a chart recorder continuously monitoring the temperature of the sensors. The part remained in the chamber at 135°C until 5 min after the internal temperature sensor stabilized at 135 ±1°C (heat-soak period[1]). The part was then removed from the chamber and exposed to the 25°C ambient room temperature, with the chart recorders continuing to monitor the sensor temperatures until 5 min after the internal sensor stabilized at 25 ±1°C. Thereby, the time lag for the interior of the part to reach the ambient (135°C simulated heat sterilization) temperature was recorded. In addition, the time for the interior of the part to return to the 25°C ambient room temperature was recorded.

d. Part type selection. The part types (Table 1) were selected from those undergoing qualification testing for inclusion in the JPL Sterilized Preferred Parts List. They were selected as representing a cross section of generic part types that would represent worst-case conditions; i.e., largest mass or containing considerable dead air space (Table 2). Two each of the five part types were tested in order to correlate and verify measurement accuracy. In addition, spare units were used experimentally to determine empirically the optimum sensor location.

e. Temperature sensor location. Spare units of each part type were physically sectioned in order to evaluate the internal temperature sensor location. In the case of the capacitor and inductor it was determined to locate the sensor in the physical center of the part. These parts were solid with no dead air space (cavities), and the

[1]The heat-soak period ensured that the part completely stabilized at the temperature.

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Table 1. Description of test items

Part	Vendor	Type No.	Value	Test sample No.
Capacitor	Sprague Electric Co.	195P10592S4	1.0 μf, 200 VDC	1,2
Relay	General Electric Co.	3SAM1301A2	30 VDC, SPDT	3,4
Inductor	MCE Corp.	EA37RT30	115 v, 400 cps	5,6
Trimming resistor	Bourns, Inc.	224L-1-503	50 K, 1.0 w	7,8
Transistor	Texas Instruments, Inc.	2N1937	20 A, 150 w	9,10

Table 2. Dimensions and weight (approximate)

Part	Dimensions, in.	Weight, oz
Capacitor	2.5 × 0.75 D	1.3
Relay	0.9 × 0.6 × 0.4	0.6
Inductor	1.60 × 1.32 × 1.28	4.2
Trimming resistor	1.25 × 0.31 × 0.19	0.1
Transistor	Case size P	0.9

center of the part was considered as being the last probable location to stabilize at the 135°C external ambient temperature. The transistor, relay, and trimming resistor had dead air space (internal cavities). Consequently, it was determined to locate the sensor, suspended in the dead air space, as far removed from the case or internal material as practicable. Suspending the sensor in the dead air space was to preclude the conduction of heat from the exterior through the internal material and directly heating the sensor. In each case, the location of the external sensor was against the largest external mass of the part. The external sensor was included for comparison purposes, but was not considered essential to the objectives of the program. The desirability of the sensor location was confirmed by monitoring the sensors during experimental test simulation. Each part was mounted on a separate glass epoxy circuit board in order to facilitate insertion and removal from the temperature chamber, and permit electrical contact with the temperature sensors.

f. Temperature measurements. The internal and external temperatures of the part were measured, using microminiature calibrated glass-bead thermistors 0.04 in. in diameter. The accuracy of the thermistors was to within ±1.0°C, and they were initially calibrated against precision thermometers accurate to 0.1°C. In the temperature gradient runs, a constant current of 0.5 ma DC was maintained through the thermistor. This current was chosen to minimize the internal heating effects of the thermistor. The voltage drop was monitored and amplified to equal the resistance of the thermistor, and this resistance value was continuously recorded on a chart recorder. The resistance of the thermistors was verified with a digital voltmeter throughout all temperature gradient runs, and the chamber temperature was monitored with calibrated thermometers accurate to 0.2°C. The thermistors selected were as small as practicable so as to permit insertion into the part with a minimum of part destruction. The board and part were inserted into the chamber through a small porthole in less than 3 sec. Consequently, the mean chamber temperature remained within ±1°C during the temperature runs.

g. Test method. Two each of each of the five part types outlined in Table 1 were individually subjected to the following tests while the temperature sensors were being continuously monitored by a chart recorder:

(1) The part was stabilized at 25 ±2°C for 5 min.

(2) The part was inserted into the 135 ±1°C preheated temperature chambers.

(3) After the internal part sensor had stabilized at 135 ±1°C, the part was heat soaked at 135 ±1°C for an additional 5 min.

(4) The part was removed from the temperature chamber.

(5) The part remained at room ambient temperature until the internal sensor stabilized at 25 ±1°C.

(6) After the internal sensor had stabilized at 25 ±1°C, the temperature was recorded for an additional 5 min.

h. Test results. The measurements from the chart recorder were transcribed onto graphs in order to facilitate interpretation. Fig. 1 is a summary of the results.

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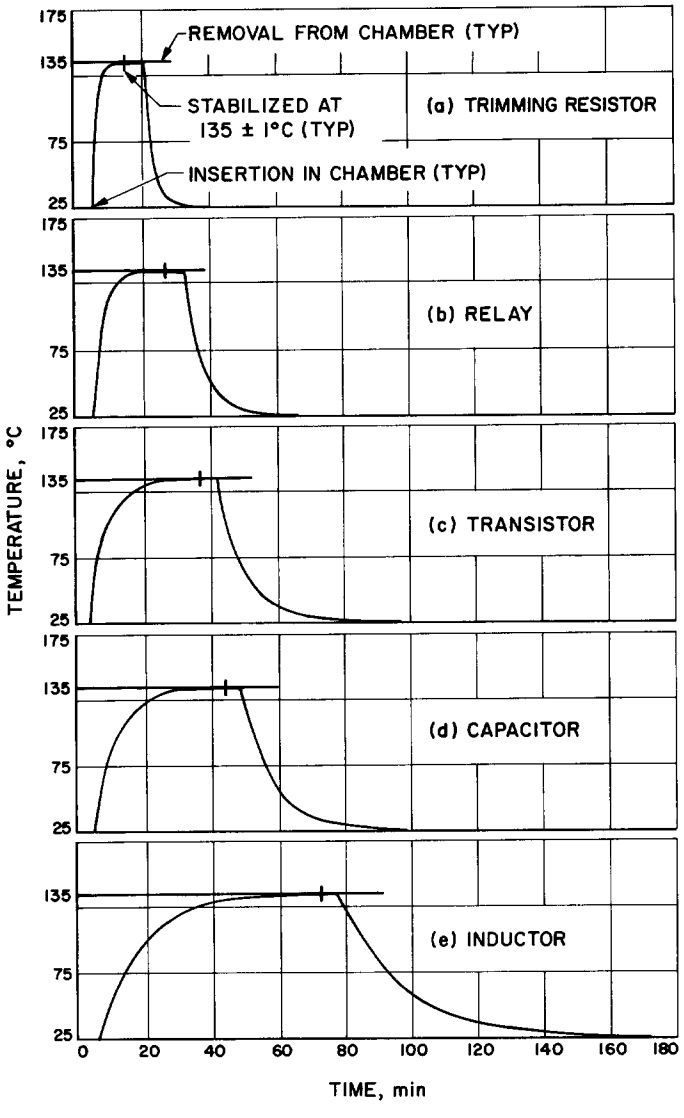

Fig. 1. Time-temperature graphs of test samples

A summary for the test runs on the various component parts are shown graphically in Fig. 1(a) through (e), and they are tabulated in Tables 3 and 4.

Table 3 outlines the total time cycle from insertion in the chamber, until removal and return of the internal sensor to 25 ±1°C room ambient temperature. This includes the 5-min heat soak period at 135 ±1°C.

Table 4 outlines the time for the internal sensor to stabilize at 135 ±1°C after insertion in the chamber. These times do not include the 5-min soak period at 135 ±1°C. The time for the internal sensor to return to 25°C after removal from the chamber is included for reference.

Table 3. Insertion cycle time

Part type	Time, min[a]	Figure No.
Trimming resistor	30 to 48	1(a)
Relay	57 to 60	1(b)
Transistor	85 to 88	1(c)
Capacitor	85 to 100	1(d)
Inductor	162 to 165	1(e)

[a]Approximate time limits for temperature stabilization.

Table 4. Time cycles for stabilization at 135°C

Type part	Time, min[a]		Figure No.
	Temperature, °C		
	25 to 135	135 to 25	
Trimming resistor	10 to 12	13 to 18	1(a)
Relay	17 to 18	29 to 32	1(b)
Transistor	30 to 32	49 to 50	1(c)
Capacitor	34 to 39	45 to 48	1(d)
Inductor	68 to 73	85 to 90	1(e)

[a]Approximate time limits for temperature stabilization.

i. Conclusions. The temperature gradient test program was intended as an exploratory effort to determine if there is a problem area with the type of component parts that were anticipated for use in the *Voyager* capsule. The results of the program indicate that there is not a problem with the type of component parts tested. Consequently, no further component part temperature gradient testing is planned at this time. The longest time required for the interior of a part to reach the 135°C heat-sterilization temperature was slightly over 1 hr (inductors). This is not considered significant, in view of the total heat sterilization cycle (92 hr at 135°C) presently required. The time recorded for the interior of the part to attain the 135°C ambient temperature is probably slightly less than would be experienced in actual heat-sterilization procedures. This is due to the necessity of destroying the part seal by inserting the thermistor, and consequently changing the thermal characteristics of the part. As small a hole as practicable (0.04 in.) was used when inserting the thermistor, and subsequently the hole was sealed with Duc-Seal insulation material, thereby minimizing the difference.

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Although the results of the test program did not indicate a significant problem with small component parts, there could be a severe problem at the system level. It required over 1 hr for the interior of a 4.2-oz inductor to reach the sterilization temperature, and considering the size and mass of the capsule, the problem becomes evident.

C. Guidance and Control

1. Spacecraft Power: Sterilizable battery

a. Development of separators. The development of a separator material capable of functioning satisfactorily in the Ag-Zn battery after heat sterilization is continuing. The two materials derived from Contract 951015 with RAI Research Corp. are being evaluated. Four other companies, the Southwest Research Institute, Lockheed-Georgia Company, Martin-Marietta Corp., and High Energy Processing Corp., are under contract to produce a uniform product, using a procedure developed under the RAI contract. The Narmco Division, Whittaker Corp., is modifying one polymeric material, the film prepared from poly-2,2′-(octamethylene)-5,5′-bibenzimidazole, to lower the resistivity value.

Materials from JPL Contract 951015. The Electric Storage Battery Company and JPL are evaluating two materials produced under this contract. In preliminary tests these materials had been shown to have the requisites of chemical and thermal stability and low resistivity. The present evaluations are to determine the uniformity of the films and their performance characteristics in Ag-Zn cells. The JPL resistivity data, which determine a degree of uniformity, are given in Table 5. RAI-116 was precrosslinked by irradiation with divinylbenzene prior to grafting with acrylic acid; RAI-110 was precrosslinked by irradiation with an electron beam prior to grafting with acrylic acid. Measurements of the electrical capacity of Ag-Zn cells built using RAI-116 as the separator are continuing; no conclusions can be made yet.

Table 5. Resistivity measurements to determine uniformity

	RAI-116	RAI-110
Before sterilization		
Number of measurements	90	36
Resistivity (arith. mean), ohm-in	18.1	54.3
Standard deviation, % of mean	50	21
After sterilization		
Number of measurements	84	18
Resistivity (arith. mean), ohm-in	5.4	—
Standard deviation, % of mean	46	—

The material produced under Contract 951015 did not have the high uniformity necessary for the separator for the sterilizable battery. To obtain reliability, four contracts were given to fabricate the RAI-116 material with the purpose of selecting a company which can produce the most uniform film. The products of these four companies—the Southwest Research Institute, Lockheed-Georgia Co., Martin-Marietta Corp., and High Energy Processing Corp.—will be evaluated by JPL, to select the one which will produce large quantities for extensive cell testing.

JPL Contract 951091 with Narmco Division, Whittaker Corp. Modifications were made to improve the resistivity value of a film fabricated from poly-2,2′-(octamethylene)-5,5′-bibenzimidazole. Hydroxyethylation and cyanoethylation of the imidazole ring and physical changes by a leaching technique with LiCl lowered the resistivity values. The most successful method was cyanoethylation, the resistivity value being reduced from 1.7×10^6 to 1 ohm-cm^2. Since the acceptance value is 0.375 ohm-cm^2, efforts to improve this polymer by further cyanoethylation are being continued.

b. Research and development. This contract covers the research and development of scaled Ag-Zn and Ag-Cd cells capable of satisfactory performance after heat sterilization in the uncharged condition and after high impact. Advances have been reported in the electrochemistry, cell case, and cell fabrication and test tasks.

There are several phases in the electrochemistry task. The conductivity of aqueous KOH solutions was measured at 10, 25, and 55°C as a function of concentration. The solubility of ZnO in aqueous KOH is being measured as a function of temperature and KOH concentration. Studies of the efficiency of additives for preventing Zn dendrite growth have shown that cellophane resists penetration better than the RAI separator. From studies of the effects of the sterilization procedure, it has been concluded that polypropylene is chemically attacked and is unsuitable for use. Tentative conclusions from data on the sterilization effects on Ag, Zn, and Cd electrodes are that: (1) Cd electrodes behave nearly the same before and after sterilization, (2) charge acceptance of sterilized

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cells made with Zn electrodes was low on the first cycle, (3) the discharge efficiency was not greatly affected by sterilization, and (4) the two layers of RAI-116 separator materials are insufficient for four electrical cycles. RAI-116 separator material testing showed that the film is not homogeneous, that electrical resistance decreases after sterilization, and the percent of electrolyte solution retained decreases by a factor of about two after sterilization.

In the cell case task, tests for chemical and thermal stability have shown that polyphenylene oxide and polysulfone are suitable polymeric materials. Cell cases have been molded from these materials and exhibit satisfactory dimensional stability after annealing. No satisfactory sealing procedure has been developed, and experiments with the techniques of ultrasonics, heating, gas welding, solvent, and adhesives are being investigated. The effort in this task is concentrated now on the sealing problem, and some initial promise of the use of welding for polysulfone cases and adhesive seals for polyphenylene oxide cases is being extensively explored.

In the cell fabrication and test task preliminary designs have been formulated and reviewed. Three-cell monoblocks have been molded from both polysulfone and polyphenylene oxide to test molding feasibility and to provide cases for high-impact testing. Designs to develop reinforced plates are under way.

2. Central Computer and Sequencer

The logical design and the programming for the central computer and sequencer (CC&S) described in *SPS 37-36*, Vol. II, p. 24, have been continued. As described in that article, the CC&S can be separated into two distinct parts: the maneuver control subsystem, with its input buffer, and the remainder of the system. The remainder, which will be referred to as the memory subsystem, contains the memories, the arithmetic and control logic, output actuators, and a separate input buffer.

The memory subsystem has been designed in some detail, using two different logic implementations. The first, composed of semiconductors and magnetic cores, is shown in *SPS 37-36*, Vol. II, Fig. 1, on p. 25. A second implementation, using a class of inhibit-core logic, has also been studied. Inhibit-core logic has the advantage of a much higher ratio of magnetic cores to active semiconductors and is expected to be more reliable. The Laboratory has constructed and tested a general-purpose inhibit-core computer to demonstrate feasibility under a program supported by the NASA-OART (*SPS 37-12*, Vol. II). The inhibit-core version of the *Voyager* CC&S memory unit uses 121 transistors, as compared with 223 for the other logical implementation.

Reliability of mission accomplishment has been a prime goal in the *Voyager* CC&S development. All CC&S functions have been divided into two groups, critical and noncritical, according to whether or not their failure would prevent accomplishment of the mission. In making this categorization, it has been assumed that only a single failure is to be guarded against, and that either the CC&S will be working properly, or a radio commanded backup system will be available. Thus the group of critical functions comprises: (1) those whose premature occurrence would cause some event which cannot later be undone through radio command, and (2) those whose delayed occurrence will prevent some event whose timing is so critical that radio command cannot be used. Most of these critical functions occur in connection with the trajectory correction maneuvers, and it is for this reason that the separate maneuver-control subsystem exists. Methods are being investigated to make this unit redundant, so that it will be unaffected by single failures. Both triple modular redundancy and a switching redundancy using the memory subsystem as a backup are being considered.

A preliminary power estimate of the complete CC&S indicates that its weight will be 47 lb, and it will consume 17 w. The estimate is based on the inhibit-core mechanization of the memory subsystem, with a maximal set of 128 output actuators. This number of actuators is not expected to be necessary in the *Voyager* mission. The estimate is also based on a triply redundant maneuver subsystem. The power estimate is the maximum required during maneuvers. During normal cruise the average power will be about 1 w.

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III. Future Projects

A. Introduction

The Future Projects Office of the Laboratory is responsible for the direction, sponsorship and, in cooperation with other divisions of the Laboratory, the origination of advanced mission and future project studies of interest and applicability to the Laboratory.

The objective of this effort is to insure that the preliminary technical information and plans for the various potential missions, together with the associated information on spacecraft systems, instrumentation systems, launching vehicle systems, resources, and schedule be available as required. This preliminary mission study information will be used primarily for defining specific flight mission programs and as generalized guidance for the Laboratory research and advanced development programs.

Each study conducted by this Future Projects Office has been, and will continue to be, documented in a formal final report. This section of the Space Programs Summary is allocated to provide an abstract or summary of the results of each such study including, where appropriate, reference to the final report for additional details.

B. Science Planning for Advanced Venus Mission Studies[1]

The two principal objectives of planetary science are the understanding of: (1) the origin and evolution of life, and (2) the origin and evolution of the solar system. In spite of the alleged high surface temperatures, Venus is still interesting from the point of view of biology and the origin and evolution of life. This is because of some uncertainty regarding the interpretation of the radio emission from the planet, the possibility of elevated topography (especially at high latitudes) having lower temperatures, and the possibility that aerial life forms have evolved, suspended in the cooler portions of the atmosphere. Moreover, Venus is similar to Earth in mass, size, and density. Therefore, it is possible that it has a core and that its surface is shaped by the same construction forces that act upon the Earth. Venus has a dense, meteorologically interesting atmosphere, and its study may yield important clues toward understanding the

[1]This section is a summary of "Venus: Preliminary Science Objectives and Experiments for Use in Advanced Mission Studies," Technical Memorandum No. 33-282, Jet Propulsion Laboratory, Pasadena, California (in process).

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Table 1. Typical Venus flyby experiments

Instrument	Weight, lb	Power, w	Design objectives
1. Microwave spectrometer (atmospheric sensor)	35	30	To obtain the limb-to-limb brightness temperature at several wavelengths from 3 mm to 3 cm, to get information about the composition, the three-dimensional physical and thermal structure, the circulation, the density, etc., of the atmosphere.
2. Microwave imager (surface imager)	25	10	To obtain a thermal map of a traverse of the Cytherean surface at a wavelength longer than 3 cm, to get information on planetary geomorphology, thermal abnormalities, and an estimate of surface composition and structure.
3. Microwave spectrometer/ imager (combined atmospheric sensor and surface imager)	50	40	Same as 1 and 2.
4. Infrared experiment	35	20	Using Michelson interferometer technique at 1.2 to 5 μ with a spectral resolution of 0.5 cm^{-1}, to secure information about atmospheric composition, vertical temperature and nature and height of clouds.
5. Magnetometer	5	5	To investigate planetary and interplanetary magnetic fields, their relationship, characteristics, magnitude, direction, and orientation.
6. Solar plasma experiment	15	10	To make detailed energy measurements, as well as flux and directional measurements similar to OGO-E plasma experiment.
7. Trapped radiation detectors	2.6	0.7	Planetary: To search for magnetically trapped charged particles in vicinity of Venus. If present, to study intensity, directional and spatial distribution. Also to measure interplanetary cosmic rays.
8. Cosmic dust	2.5	0.2	Interplanetary: To measure flux of cosmic dust particles as function of direction, distance from Sun, and momentum with respect to spacecraft.
9. RF Occultation			Planetary: To measure changes in spacecraft radio signal resulting from effect of Venus atmosphere during occultation.
10. Particle flux/ion chamber	2.5	0.35	Particle Flux, interplanetary: In conjunction with ion chamber, to monitor energetic particle and photon radiation in interplanetary space. Provides semiquantitative data about energy and particle types composing the radiation. Ion Chamber, interplanetary: To measure total ionizing radiation (cosmic rays and/or particles) and variation with time and position in the solar system.
11. Photo-imaging experiment	15	20	To provide information about nature and structure of Cytherean atmosphere similar to data from early U.S. satellites for Earth.
12. Ultraviolet spectroscopy	25	12	To determine abundance and height distributions of principal atmospheric constituents obtained from UV spectra of day, night, and twilight portions of Venus, with emphasis on lighted side.
13. Radar experiment	50	75	To provide information on surface profile or topography, range to planetary surface, small-scale surface roughness (1 to 30 cm), and reflectivity and electrical properties of planetary surface.

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origin and evolution of planetary atmospheres. The study of Venus thus can be very important in a better understanding of terrestrial problems, as well as providing information useful for a solution of the two principal objectives of planetary science.

A launch vehicle with at least the capabilities of an *Atlas/Centaur* can be assumed for this mission. This means a *Mariner*-type spacecraft weighing 1,500 lb or more. This order of capability suggests the possibility of more than a simple flyby mission; rather, a preliminary analysis indicates that perhaps three types of missions are practical: (1) flyby only, (2) Venus capsule only, and (3) flyby/capsule combination.

The flyby-only mission is feasible in that it is a comparatively easy mission to perform, and it offers a high probability of obtaining some of the necessary Venus scientific data. However, it is extremely difficult, from a flyby spacecraft, to obtain unambiguous information on, e.g., the temperature, pressure, density, etc., of the Cytherean atmosphere as a function of altitude, or e.g., the hardness of the planetary surface. At the same time, the capsule-only mission omits the planetary measurements that can be made from the flyby spacecraft concerning cloud structure, surface structure, and composition of the upper atmosphere. It is also difficult to perform a capsule-only mission with the many environmental and mechanization uncertainties reducing the probability of mission success.

Of the three missions, the flyby/capsule combination is the most interesting, since it has the potential for providing the highest return in scientific information. It not only permits direct measurement of the temperature, pressure, and density of the Cytherean atmosphere as a function of altitude, and perhaps an order-of-magnitude measurement of the hardness of the surface, but it also permits radar and microwave examination of the planetary surface, as well as ultraviolet and infrared examination of the upper atmosphere. These factors, together with a certain amount of redundancy of measurements between the flyby spacecraft and the capsule, make this mission of keen interest scientifically.

Scientific investigations that can be conducted from the flyby spacecraft involve both planetary and interplanetary experiments. The planetary experiments involve the measurement of select portions of the electromagnetic spectrum, ranging from the vacuum ultraviolet and far infrared to microwave and radar frequencies. The instruments can vary from reasonably heavy, complicated, high-data-rate spectrometers, interferometers, image-forming and scanning devices to relatively simple, lightweight radiometers and photometers. They all have narrow fields of view and exacting pointing and aiming requirements relative to the planet. It is possible that several instruments could be mounted on a common scan platform to provide the gross aiming capability. A Venus flyby miss distance as close as 1000 km is desirable for these experiments; however, a miss distance of up to 5000 km would be satisfactory.

The interplanetary experiments involve the measurement of planetary magnetic fields and trapped radiations, as well as the interplanetary measurements. The several experiments considered here in this category have either been flown before, or their application to this mission involves a relatively straightforward extension of instrument state of the art. Table 1 lists some possible scientific instruments for flyby science.

A simple capsule or drop sonde can be instrumented to provide first-hand information on the temperature,

Table 2. Typical capsule instruments

	Weight	Power, w
Instruments to measure thermodynamic variables:	3 lb	2.5
Temperature	5 oz	
Pressure	6 oz	
Density	24 oz	
Velocity of sound	10 oz	
Instruments to measure atmospheric composition (only one from the following categories will be selected):	7 lb	6
Mass spectrometer	5 lb	6
Gas chromatograph	6.5 lb	5
Simple composition	8 lb	6
H_2O	1.5 lb	1
O_2	1.5 lb	1
O_3	1.5 lb	1
A	1.5 lb	1
N_2	1.0 lb	1
CO_2	1.0 lb	1
Visual photometer	2 lb	1
Impact accelerometer	3 lb	1
Science data system	4 lb	2
Totals	19 lb	12.5

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pressure, density, and some aspects of the chemical composition of the Cytherean atmosphere, all as a function of altitude. A sonde of this type, although not designed to survive the ballistic impact on the planet's surface, should carry some kind of accelerometer to distinguish between solid and liquid surfaces. Table 2 lists some possible scientific instruments to carry out each of these measurement objectives.

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